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NASA Technical Memorandum 78641

NUMERICAL DESIGN OF STREAMLINED TUNNEL WALLS FOR A TWO-DIMENSIONAL TRANSONIC TEST

(NASA-TM-78641) NUMERICAL DESIGN OF
STREAMLINED TUNNEL WALLS FOR A
TWO-DIMENSIONAL TRANSONIC TEST (NASA)
HC A02/MF A01

24 p
CSCL 14B

N78-23105

G3/09 16761
Unclass

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April 1978



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SUMMARY

An analytical procedure is discussed for designing wall shapes for streamlined, nonporous, two-dimensional, transonic wind tunnels. It is based upon currently available 2-D inviscid transonic and boundary-layer analysis computer programs. Predicted wall shapes are compared with experimental data obtained from the NASA Langley 6- by 19-Inch Transonic Tunnel where the slotted walls were replaced by flexible nonporous walls. Comparisons are presented for the empty tunnel operating at a Mach number of 0.9 and for a supercritical test of an NACA 0012 airfoil at zero lift. Satisfactory agreement is obtained between the analytically and experimentally determined wall shapes.

INTRODUCTION

The present procedure for determining streamlined wind-tunnel liner shapes is being developed to design a contoured nonporous liner for simulating free-air transonic flow about an infinite aspect-ratio yawed wing. Such a liner will be required for testing a large-chord, laminar flow control (LFC), swept-wing panel, which has a supercritical airfoil section, in the NASA Ames 12-Foot Pressure Wind Tunnel at supercritical flow conditions. The nonporous liner shape is to be defined by a number of streamlines which have been calculated for free-air flow about an infinite aspect-ratio yawed wing and which are further displaced according to a calculated boundary-layer displacement thickness in order to account for the blockage of the viscous layer on the real wall.

This numerical procedure is applicable to a two-dimensional tunnel and a first check of its validity is provided by a direct comparison of the analytically-determined tunnel wall shape with one determined experimentally in a streamlined tunnel experiment. Barnwell and Everhart* have recently completed such an experiment for a symmetric airfoil at zero lift in the NASA Langley 6- by 19-Inch Transonic Tunnel. For that test, the slotted walls of the tunnel were replaced by nonporous flexible plates which were adjusted during the experiment in order to simulate free-air flow over the airfoil section. Thus, the tunnel-wall displacement required to achieve equivalent free-air flow is determined as a result of the experiment. Their iterative experimental/analytical technique is similar to that presented by Goodyer (ref. 1).

The purpose of this paper is to show that use of currently available numerical analysis procedures gives meaningful results which can be used in two-dimensional adaptive-wall tunnels operating in the transonic (supercritical) flow regime. The sections which follow describe the present method as it pertains to two-dimensional tunnel applications; very briefly outline the Barnwell-Everhart experiment; and, finally, compare results for tunnel wall shapes. The present procedure has, of course, a general utility which is restricted by our current ability to calculate the viscous flow field about arbitrary configurations. However, continuing rapid advances in computational machinery and methods will allow for better numerical modeling of the more complex phenomena associated with strong shocks and high lift.

* A detailed description of the streamlining procedure and results obtained from it have not yet been published by R. W. Barnwell and J. L. Everhart of NASA Langley Research Center. The present analytical results are compared with their experimentally determined wall shape; we appreciate and acknowledge this early release of their data. In this Paper their procedure and results will be identified as Barnwell-Everhart.

DESCRIPTION OF METHOD

The present numerical method for determining contoured-wall shapes which minimize wind-tunnel interference at transonic test conditions for large models is a direct application of several existing numerical techniques. First, a transonic analysis code is used to calculate the flow field about the effective inviscid model shape; that is, the geometric model plus a boundary-layer correction on the model. Most two-dimensional airfoil codes routinely used throughout the aerospace industry (refs. 2 and 3, for example) perform this task. In order to determine free-air streamline locations and pressures along them, one must use results which were computed throughout the flow field. When one replaces the bounding free-air streamlines with a contoured tunnel wall, then some account must be made for the resulting wall boundary layer. In the present method, a two-dimensional boundary-layer code (ref. 4) is used to calculate a displacement correction along each streamline which is to be replaced by a wall. The edge condition for this calculation is determined from the local pressures calculated for the flow field about the model. For nonporous contoured walls at transonic test conditions, this viscous blockage correction must be made in order to prevent the streamlined tunnel from choking. This is the basic idea of the procedure; each particular application will differ somewhat in detail. Subsequent paragraphs indicate what was done in order to model the Barnwell-Everhart experiment.

The symmetric airfoil used by Barnwell and Everhart was an NACA 0012 at zero incidence at the freestream Mach number 0.765. Under these conditions, there is a small supersonic region terminated by a weak shockwave. When strong shocks are expected to occur, one must use a transonic airfoil analysis code which is conservative in order to predict the proper streamline locations (ref. 5). However, those used for the present application were nonconservative; thus, there is the possibility that a very small mass increase could occur at the weak shock and disrupt the

deduced in order to obtain uniform pressures through the test section. This first wall correction was intended to get a large part of the wall movement accomplished before putting the model in place. With the model in place, four successive wall corrections were made until it was deemed that the flow simulating the desired free-air condition had been achieved within the accuracies of the pressure measurement and wall-jack adjustment.

COMPARISON OF RESULTS

In this section, results obtained using the present analytical procedure for determining the flexible tunnel-wall shape are compared with those obtained experimentally by Barnwell and Everhart. For convenience in presenting the data, all dimensions are normalized with respect to the airfoil chord length, c . Two sets of experimental data have been obtained. The first set of experiments was made to determine the contour of the flexible walls which resulted in a uniform pressure in the test section of the empty tunnel. The second set of experiments was made to define the flexible-wall contour for testing an NACA-0012 symmetric airfoil at zero incidence with minimal wall interference.

Empty Tunnel Results

For the empty tunnel test, the tunnel operating conditions were:

Test section Mach number, $M_\infty = 0.902$

Test section Reynolds number, $N_{Re,\infty} = 2.24 \times 10^5 / \text{cm}$

Stagnation pressure, $P_o = 1.60 \times 10^5 \text{ Pa}$

Stagnation temperature, $T_o = 302\text{K}$

In order to define the flexible-plate contour experimentally for the empty tunnel, pressure measurements were made on both the tunnel flexible and rigid walls. Perfect gas relations were then used along with the area relation to determine the effective displaced tunnel dimensions. Since only the narrow, flexible,

4.333 chords ahead of the airfoil leading edge and the calculated free-air flow-field perturbations there are very small. If one were to neglect the wall viscous effects, these inviscid streamlines would be matched to the fixed-tunnel section and form a planar sidewall downstream of the junction; the outermost streamline would form the slightly deformed lateral wall, opposite the airfoil surfaces.

Boundary-layer displacement effects are calculated along inviscid streamlines which are to be the tunnel walls using a two-dimensional, finite-difference, laminar-turbulent, boundary-layer code (ref. 4). The boundary layer in the tunnel upstream of the fixed-wall/flexible-wall junction must be considered since it determines the starting profile at the junction. The three-dimensional contraction geometry of the NASA Langley 6- by 19-Inch Transonic Tunnel was replaced by an equivalent two-dimensional configuration and boundary-layer edge conditions were determined from the resulting streamwise area distributions. Transitional and fully-developed turbulent boundary-layer calculations were made along the surface of the curved lateral wall as well as the straight sidewall for these edge conditions. Downstream of the contraction section, differences between the four solutions were found to be negligible; the fully-turbulent solution made along the tunnel sidewall has been used as the starting solution at the junction for all viscous calculations along free-air streamlines.

This two-dimensional procedure is not expected to be very realistic for the streamlines very close to the corner defined by the intersection of the model (which spans the tunnel) and the wall. In the vicinity of this intersection, three-dimensional effects are significant. These streamlines are also those which are very close to the stagnation point at the airfoil leading edge (trailing edge also if the airfoil has a large included angle there). The adverse pressure gradient at this peak causes the sidewall boundary layer to thicken appreciably and probably to separate; however, the rapid expansion around the leading edge quickly thins it out so that the influence is locally confined.

More will be said about this aspect later. In any case, it is felt that errors in the total blockage correction at each stream-wise location are not too great unless the separation or three-dimensional buildup represents an appreciable part of the entire correction.

For a tunnel with flexible (or adaptive) walls on all sides, the local boundary-layer displacement correction, $\delta^*(x,y)$, along each bounding streamline is simply applied to relieve the blockage. In the Barnwell-Everhart experiment (as well as most other two-dimensional adaptive wall tunnels), only the two lateral walls opposite the airfoil surfaces are flexible. (See fig. 1.) Thus, at each stream wise location, x , the $\delta^*(x,y)$ must be numerically integrated around the tunnel's entire local cross-sectional perimeter (i.e., across all bounding streamlines) and then applied as a displacement of only the two narrow flexible lateral walls. In the present case, the flexible-wall displacement at each stream-wise location is composed of the sum of two parts: (1) that due to the departure of the outer bounding inviscid free-air streamline from a straight line - the compressible blockage, and (2) that due to the integrated viscous displacement correction for all bounding streamlines which are to be walls - the viscous (wall) blockage. Since the wall correction actually made by Barnwell and Everhart was zero at the upstream fixed-wall/flexible-wall junction, the analytical displacement to be compared with their experimental result must be a relative (differential) displacement with respect to that at the upstream junction.

OUTLINE OF EXPERIMENT

The technique used by Barnwell and Everhart is similar to that presented by Goodyer (ref. 1). Basically, their experimental/analytical procedure iterates on the contoured-wall shape until consistency is obtained between the near-field and far-field solutions. The near-field (interior) solution is determined experimentally by the wind tunnel; that is, for a given wall

setting one observes pressures along the flexible wall (with the model in place) at the desired free-stream conditions. An inverse far-field (exterior) solution corresponding to uniform parallel flow at ∞ is obtained numerically; that is, using the measured flexible-wall pressures one solves for the shape producing it. This process is iterated until the wall shape and pressure changes are within the experimental measurement accuracies. This procedure can be looked at as an iterative solution of coupled boundary-value problems where the direct interior problem is solved by the tunnel while the inverse exterior problem is solved numerically. Since the pressures along the flexible wall are measured in the presence of viscous effects on both the interior tunnel wall and the model, the tunnel blockage due to these effects has automatically been taken into account in their procedure.

Carnwell and Everhart used the NASA Langley 6- by 19-Inch Transonic Tunnel (ref. 6) in order to verify their experimental/analytical procedure for minimizing tunnel interference for testing at supercritical speeds. The tunnel was modified by replacing the narrow lateral slotted walls with jack-supported flexible plates in the region downstream of the contraction section. (See fig. 1.) The extent of this flexible wall section relative to the 15.24-cm (6-inch) airfoil chord length, c , is $-4.833 \leq x/c \leq 3.333$. The x -origin is the center of the model and is located in the test section as shown by the $\oplus$ on figure 1. The first step in their procedure was to determine the wall shape required to produce uniform pressure for the empty tunnel (i.e., no model). This boundary-layer correction was required because the sidewalls of the existing tunnel are parallel. Tunnel runs were made at several Mach numbers; they found that the empty tunnel correction required at $M_\infty \sim 0.9$ produced adequate results at the lower Mach numbers. For this correction, the measured pressures at the flexible wall (and also the rigid wall) were plotted and wall corrections were

deduced in order to obtain uniform pressures through the test section. This first wall correction was intended to get a large part of the wall movement accomplished before putting the model in place. With the model in place, four successive wall corrections were made until it was deemed that the flow simulating the desired free-air condition had been achieved within the accuracies of the pressure measurement and wall-jack adjustment.

COMPARISON OF RESULTS

In this section, results obtained using the present analytical procedure for determining the flexible tunnel-wall shape are compared with those obtained experimentally by Barnwell and Everhart. For convenience in presenting the data, all dimensions are normalized with respect to the airfoil chord length, c . Two sets of experimental data have been obtained. The first set of experiments was made to determine the contour of the flexible walls which resulted in a uniform pressure in the test section of the empty tunnel. The second set of experiments was made to define the flexible-wall contour for testing an NACA-0012 symmetric airfoil at zero incidence with minimal wall interference.

Empty Tunnel Results

For the empty tunnel test, the tunnel operating conditions were:

Test section Mach number, $M_\infty = 0.902$

Test section Reynolds number, $N_{Re, \infty} = 2.24 \times 10^5 / \text{cm}$

Stagnation pressure, $P_0 = 1.60 \times 10^5 \text{ Pa}$

Stagnation temperature, $T_0 = 302\text{K}$

In order to define the flexible-plate contour experimentally for the empty tunnel, pressure measurements were made on both the tunnel flexible and rigid walls. Perfect gas relations were then used along with the area relation to determine the effective displaced tunnel dimensions. Since only the narrow, flexible,

lateral walls could be adjusted, all of the displacement area correction was made symmetrically there. Successive pressure measurements and displacement corrections were made until the desired uniform pressure distribution was obtained.

In the present numerical analysis, flexible-wall contours for the empty tunnel have been determined using the same general procedure as with a model; the streamlines are, however, the existing tunnel lines. In order to facilitate the numerical analysis, the tunnel contraction section was represented by an equivalent two-dimensional configuration having a width of 15.24 cm (6 inches) with the same area distribution as that of the physical configuration. Edge conditions for boundary-layer calculations have been determined using perfect-gas relations and the tunnel area distribution. Fully-turbulent boundary-layer calculations made along the plane rigid wall of the equivalent tunnel configuration were found to adequately model the correction. In order to be consistent with the experiment, the displacement correction is presented as a relative displacement taken with respect to that at the upstream junction between the flexible and rigid walls (i.e., at $x/c = -4.833$). The coordinate, y/c , defining the flexible-wall contour is measured laterally across the rigid wall (i.e., normal to the tunnel symmetry plane which is to be the model symmetry plane) with the origin as shown by the $\oplus$ in figure 1.

Comparisons of the numerically-determined flexible-wall contour and that found experimentally by Barnwell and Everhart are shown in figure 2. The experimental displacements (denoted by the symbols) are those at the 11 wall-jack locations; intermediate values were not measured. Since the flexible wall is a thin plate, intermediate values should probably be interpolated using a spline fit. The maximum difference between the analytical- and experimental-displacement-thickness corrections is approximately 18 percent of the experimental displacement at $x/c = 0$. This difference appears to be excessive; however, it corresponds to a difference in tunnel area ratios of only 0.44 percent. Wall

pressures were measured using transducers having an accuracy of ± 1 percent of full-scale deflection which, at these empty-tunnel test conditions, is equivalent to a total area-ratio difference of 0.36 percent.

NACA 0012 Model Results

The tunnel operating conditions for the airfoil test were:

Free-stream Mach number, $M_\infty = 0.765$

Free-stream Reynolds number, $N_{Re,\infty} = 2.00 \times 10^5 / \text{cm}$

Stagnation pressure, $P_O = 1.42 \times 10^5 \text{ Pa}$

Stagnation temperature, $T_O = 282\text{K}$

The test model is an NACA 0012 airfoil at zero lift with a chord length of 15.24 cm (6 inches); it is centered on the $\oplus$ mark shown in figure 1.

The numerical procedure used in the present analysis of the airfoil experiment requires a free-air solution for the inviscid flow about the airfoil including viscous displacement effects on the model, which was obtained using the methods described in references 2 and 3. At the upstream junction between the flexible and rigid tunnel walls ($x/c = -4.833$), 48 streamlines lying between $y/c = 0$ and $y/c = 1.583$ were extracted from the free-air solution for boundary-layer analysis. Figure 3 shows four representative free-air streamline contours for flow over the airfoil. The streamlines are identified by the value of $Y(-\infty)/c$ which gives the streamline position (relative to the plane of symmetry at $y/c = 0$) in the undisturbed upstream flow. Figure 3 also shows the streamwise locations of six lateral cuts across the tunnel sidewalls where representative calculated results are presented.

Conventional two-dimensional boundary-layer equations cannot be integrated into a stagnation region with severe adverse pressure gradients, so some modification must be made along streamlines near the airfoil/tunnel-sidewall junction. The procedure

used in the present analysis was to replace four values of the pressure coefficients along each of four streamlines within $0 \leq Y(-\infty)/c < 0.056$ with the corresponding four values of pressure coefficients for the streamline at $Y(-\infty)/c = 0.061$. This is the streamwise region $-0.57 \leq x/c \leq -0.49$ at the airfoil leading edge. This procedure reduces the level of the adverse pressure gradient to a value slightly below that for a predicted boundary-layer separation. Since the pressure gradient downstream of the leading edge region where the modified pressure coefficients are used is strongly favorable, the influence upon the downstream boundary-layer solutions along these streamlines is negligible. It is noted that all of the modified streamlines are close to the airfoil/tunnel-sidewall junction where the flow is fully three dimensional. As mentioned previously, the present analysis is not expected to be accurate either with or without modifications in the vicinity of this junction.

Figure 4 presents the boundary-layer thickness distributions across the tunnel sidewall (i.e., laterally) at the six streamwise locations indicated on figure 3. The displacement thickness distributions at $x/c = -2.10$ and $x/c = -1.07$ show the upstream influence of the airfoil pressure field on the boundary layer. At upstream distances beyond $x/c \approx -3.5$, the displacement-thickness distribution with respect to y/c is essentially a constant. At $x/c = -0.57$, streamlines lying below that one at $Y(-\infty)/c = 0.13$ are in an adverse pressure gradient and those lying above it are in a favorable pressure gradient. The influence of the favorable pressure gradient is clearly shown (fig. 4(a)) in the range $0.3 < y/c < 1.2$ where the displacement thicknesses are less than those at an upstream location. The inflection point in the displacement thickness at $x/c = -0.31$ is the result of different flow acceleration rates along the streamlines, and the difference in distance along the streamlines relative to their respective upstream peak positive pressure coefficient values. It is noted that more drastic changes in the pressure coefficients than those used in the analysis (to negotiate the stagnation region) gave

essentially identical results at this airfoil location and downstream of it. The downstream locations $x/c = 0.506$ and $x/c = 1.07$ show the influence of the trailing-edge compression and a representative profile in the wake region. At the downstream location, $x/c = 1.07$, the inflection point in the displacement-thickness profile results from the more rapid flow acceleration along the streamlines closer to the corner of the blunt trailing edge of the airfoil (plus boundary layer on it). Since the inviscid solutions used do not properly account for the wake effect, these displacement correction results should not be considered too realistic in this region.

Displacement-thickness distributions along the four streamlines shown in figure 2 are presented in figure 5. The behavior of these profiles is very similar to the pressure coefficient distributions along the streamlines. In figure 6, the pressure coefficient distributions along the two streamlines $Y(-\infty)/c = 0.0456$ and $Y(-\infty)/c = 1.5833$ are shown and this similarity is readily apparent.

The total displacement of the flexible tunnel walls at each x/c location is the sum of that due to the outer free-air streamline plus that due to the viscous layer on all of the tunnel walls. The tunnel-wall-displacement correction is made as previously indicated. Comparison of the analytically-determined flexible wall shape with the Barnwell-Everhart experimental data is presented in figure 7. Again, the symbols give the experimental displacement at the 11 wall-jack locations. The maximum difference in the data is at $x/c = -0.667$. The difference there in the two wall displacement corrections is less than 8 percent and the ratio of the two coordinates is 1.002. The agreement between experimental and analytical wall contours is regarded as very good.

CONCLUDING REMARKS

The present method for determining the contours of flexible nonporous wind-tunnel walls has been found adequate for the test cases considered. However, it is not immediately clear why the agreement with experimental data is better for the more complex case of an airfoil test than for the simple empty-tunnel test. One possible explanation is that the experimental data were determined using a more complex procedure for the airfoil experiment than that used for the empty-tunnel experiment.

The principal limitations of the method are due to our present inability to calculate certain transonic flow phenomena. These are:

- (a) The displacement correction on the airfoil surface does not properly account for the viscous influence in the airfoil wake.

- (b) The three-dimensional viscous flow at the junction of an airfoil and the tunnel sidewalls is not accounted for.

- (c) The procedure cannot be applied to the analysis of airfoils having a large region of separated flow.

- (d) An inviscid transonic code in conservation form should be used when strong shockwaves are to be expected.

The principal conclusions which can be drawn from these limited comparisons are:

- (a) The agreement with Barnwell-Everhart experimental results provides some degree of verification of the present numerical procedure for liner design.

- (b) The detailed streamwise resolution (or control) required at the lateral adaptive wall is seen to be caused by the response of the sidewall boundary layer to the pressure field of the model. This is, of course, most pronounced near the model.

- (c) The blockage correction due to the sidewall viscous layer cannot be neglected in streamlining nonporous tunnel walls.

(d) Blockage corrections alone will not necessarily account for the local viscous displacement effects which, at more extreme conditions, can modify the desired flow. That is, there will be local 3-D effects in the 2-D tunnel.

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NASA LANGLEY 6- BY 19-INCH TRANSONIC TUNNEL WITH SLOTTED WALLS REPLACED BY FLEXIBLE WALLS

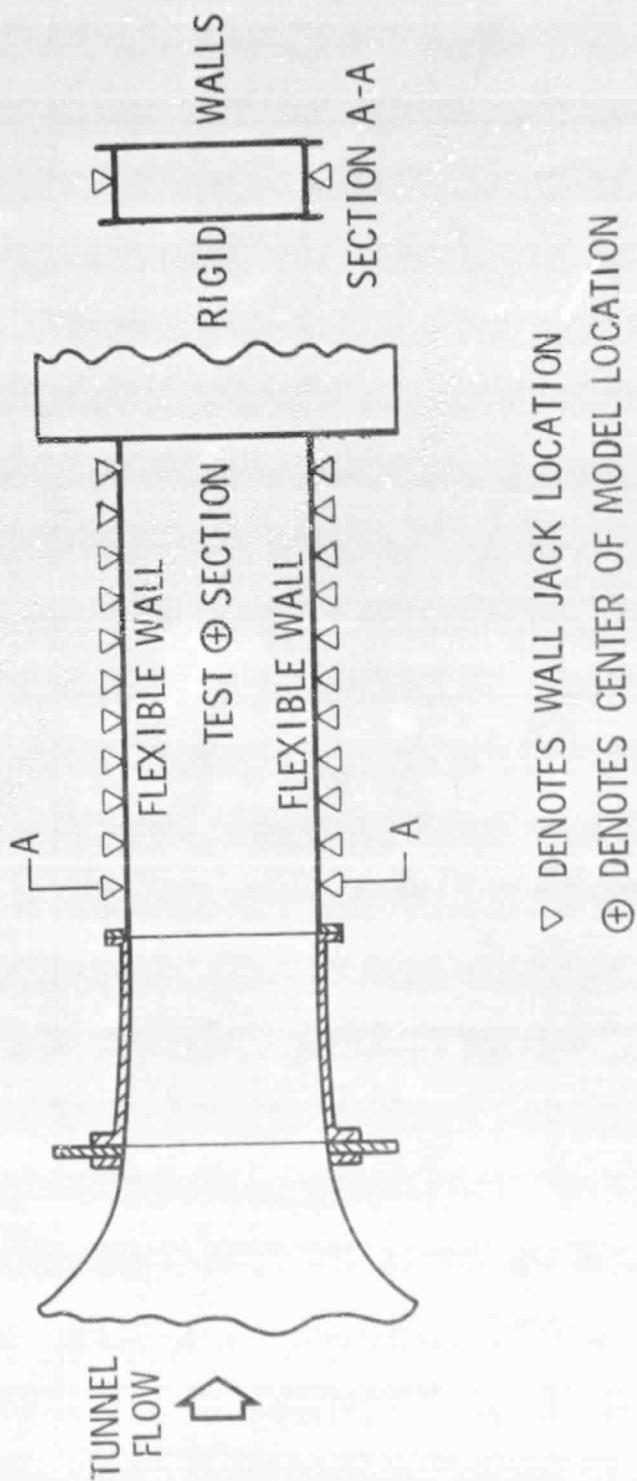


Figure 1.- Schematic of two-dimensional "streamlined tunnel" experiment of Barnwell and Everhart.

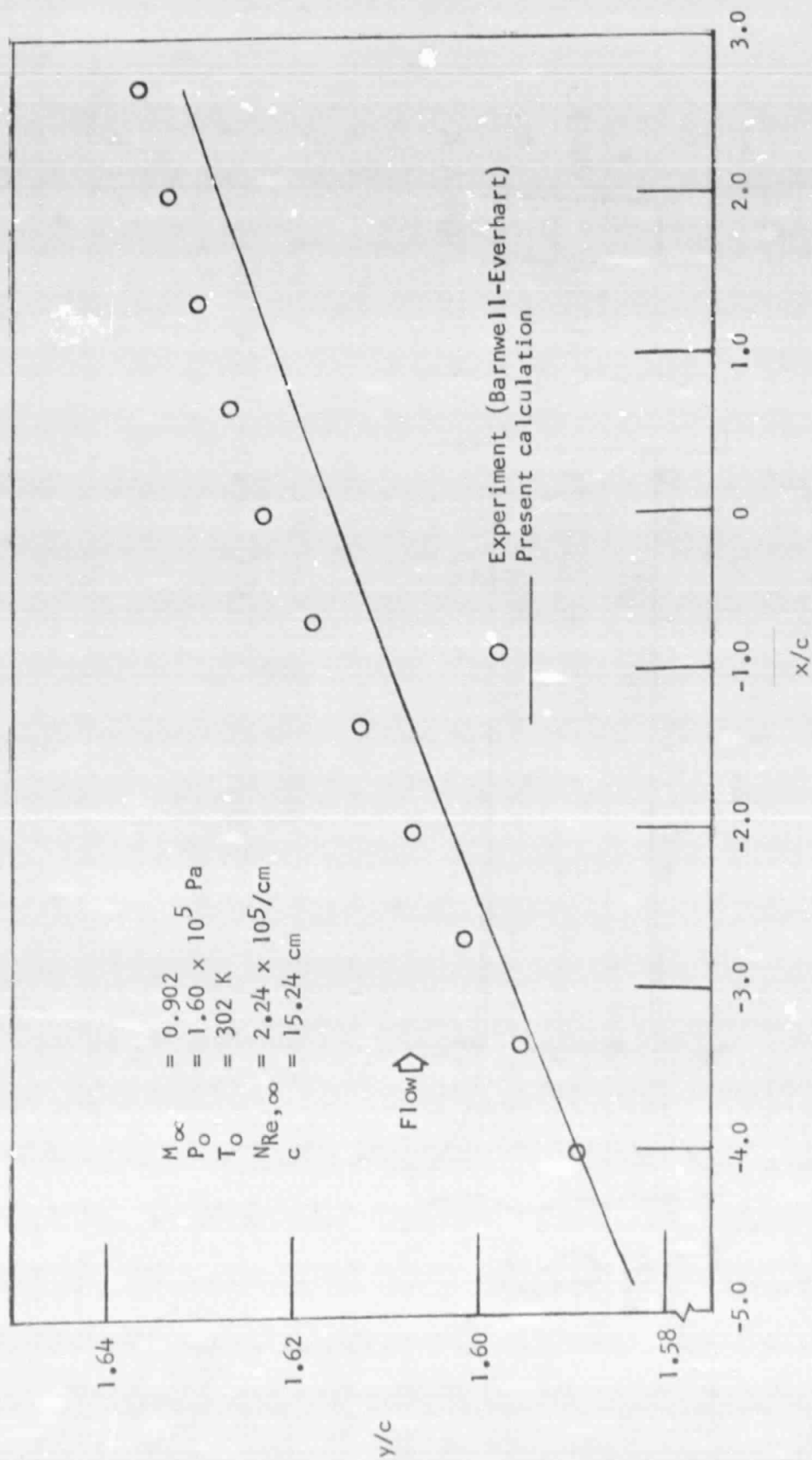


Figure 2.- Comparison of flexible nonporous wall shapes required to produce uniform parallel flow in the NASA Langley 6- by 19-Inch Transonic Tunnel.

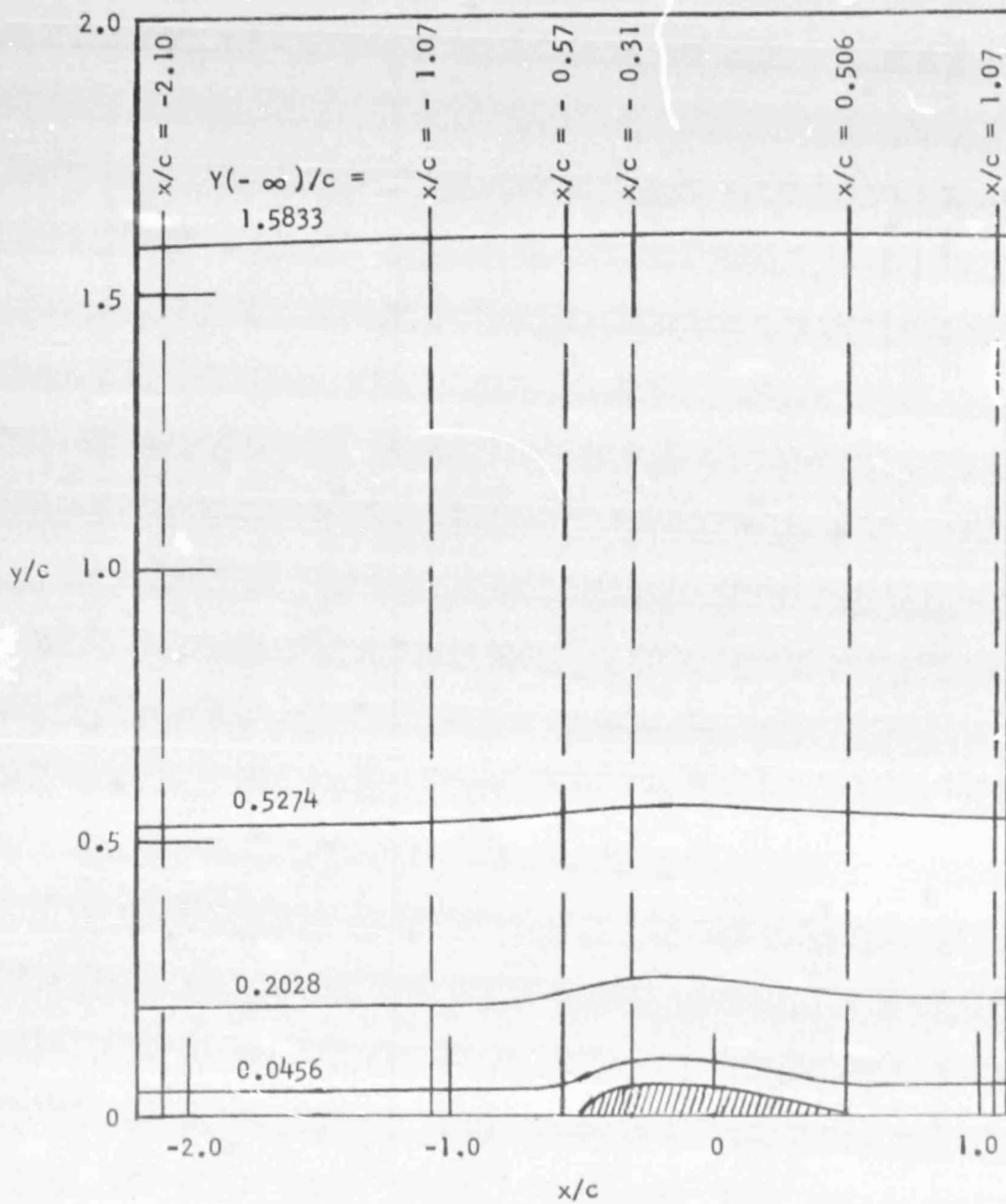


Figure 3.- Physical location of four streamlines and six lateral cuts across the tunnel sidewall where representative calculated results are given.

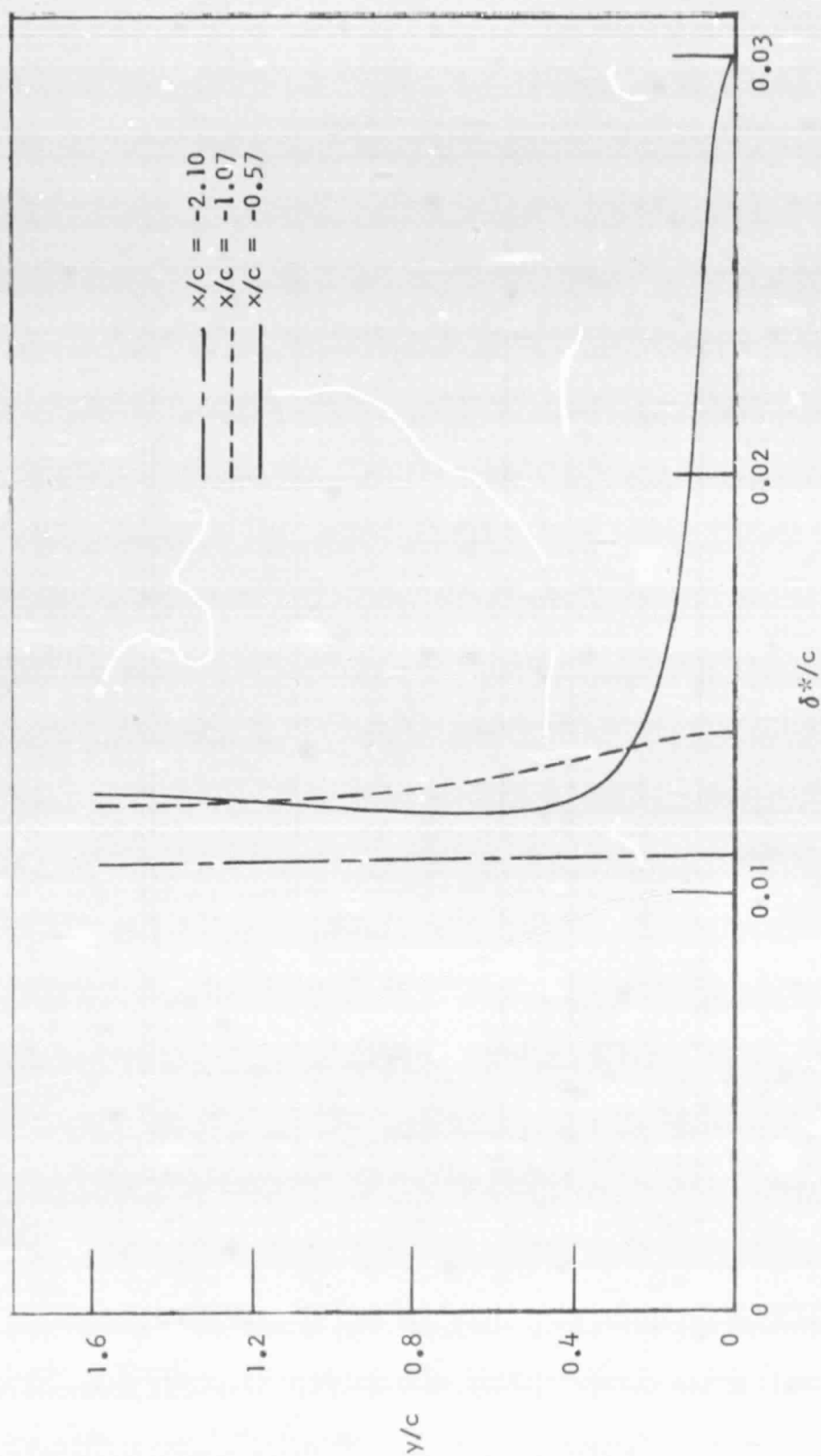


Figure 4a.- Calculated boundary layer displacement thickness distributions on the tunnel sidewall across six lateral cuts.

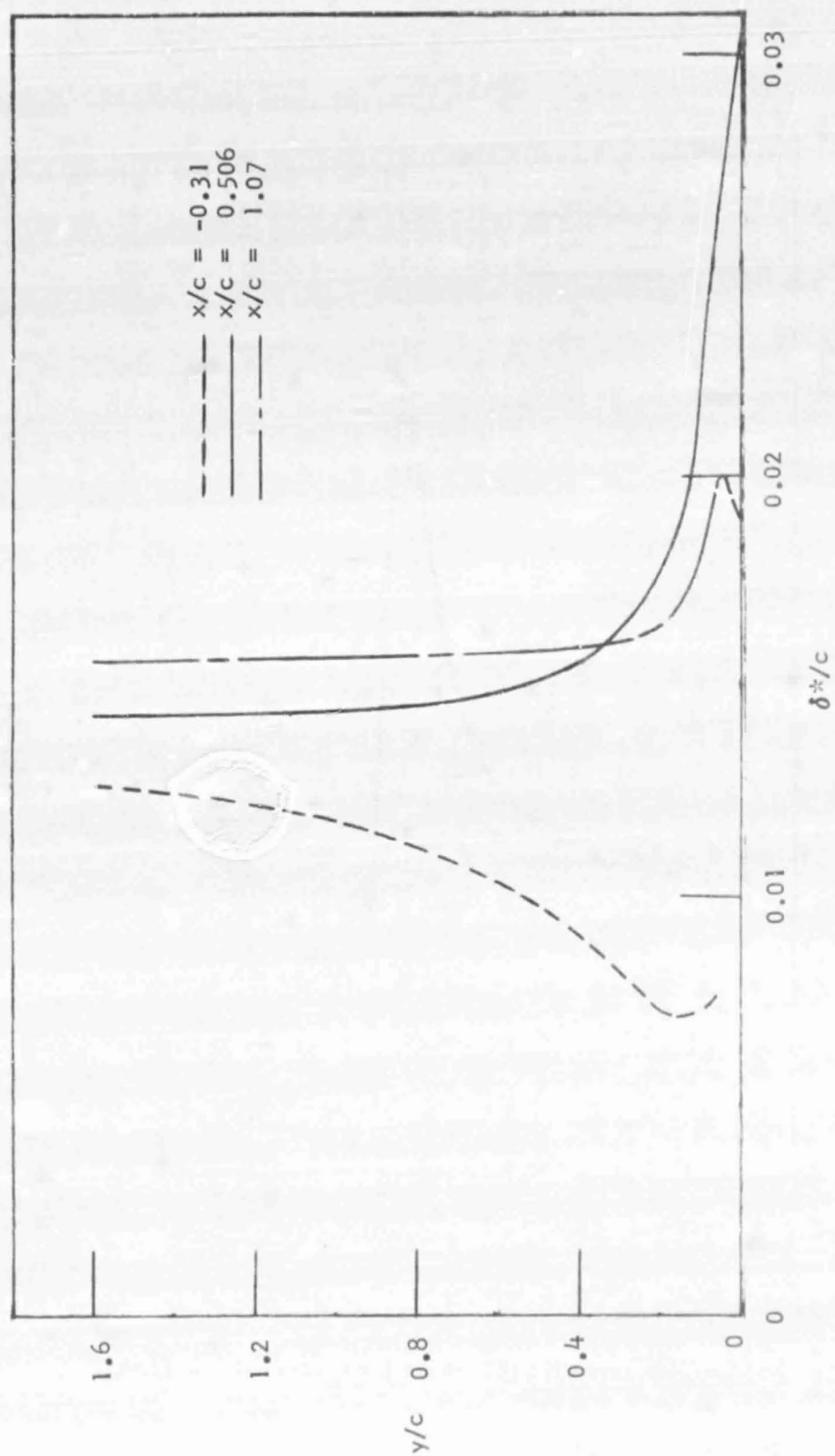


Figure 4b.- Concluded.

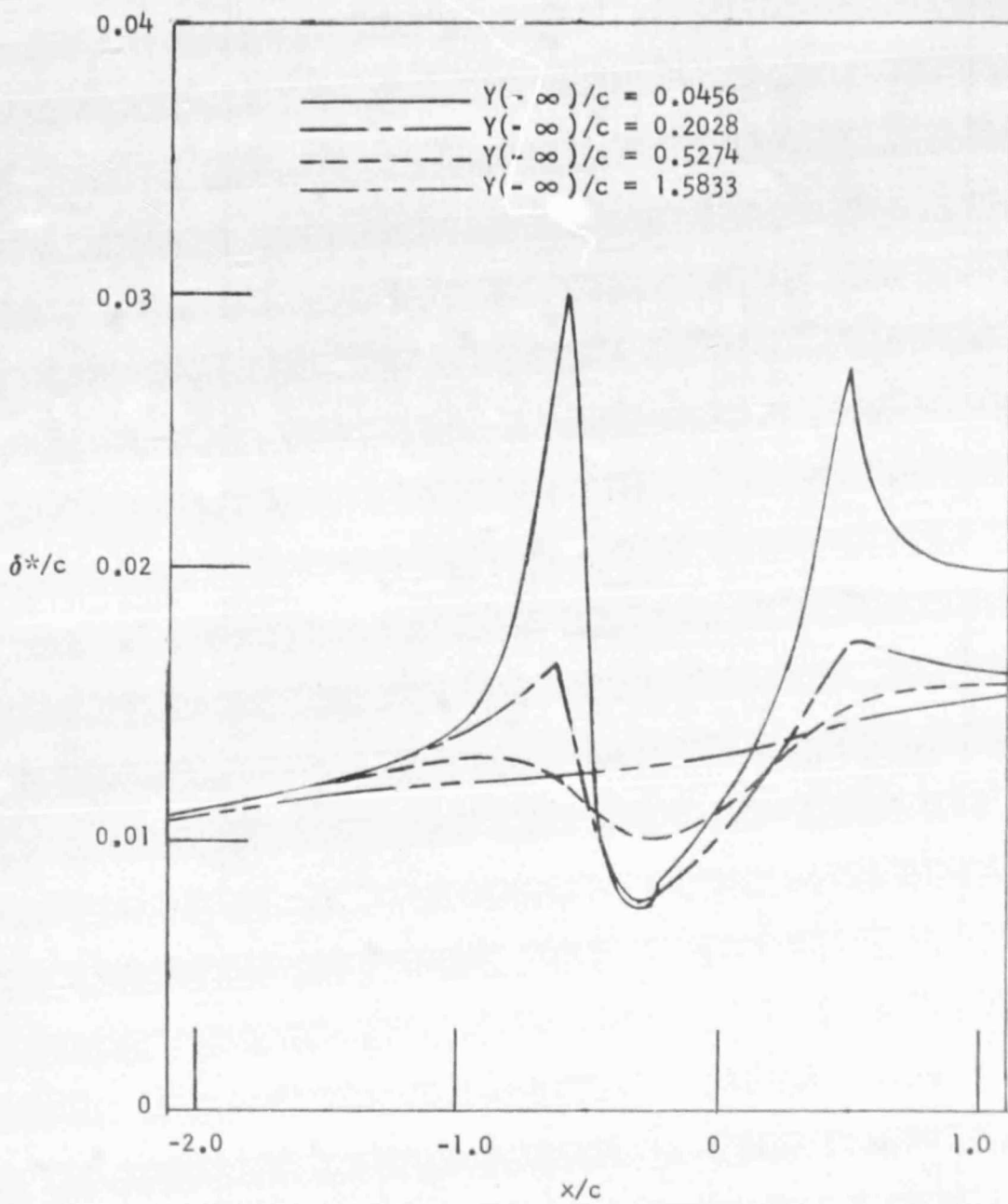


Figure 5.- Calculated boundary layer displacement thickness distributions on the tunnel sidewall along four streamlines.

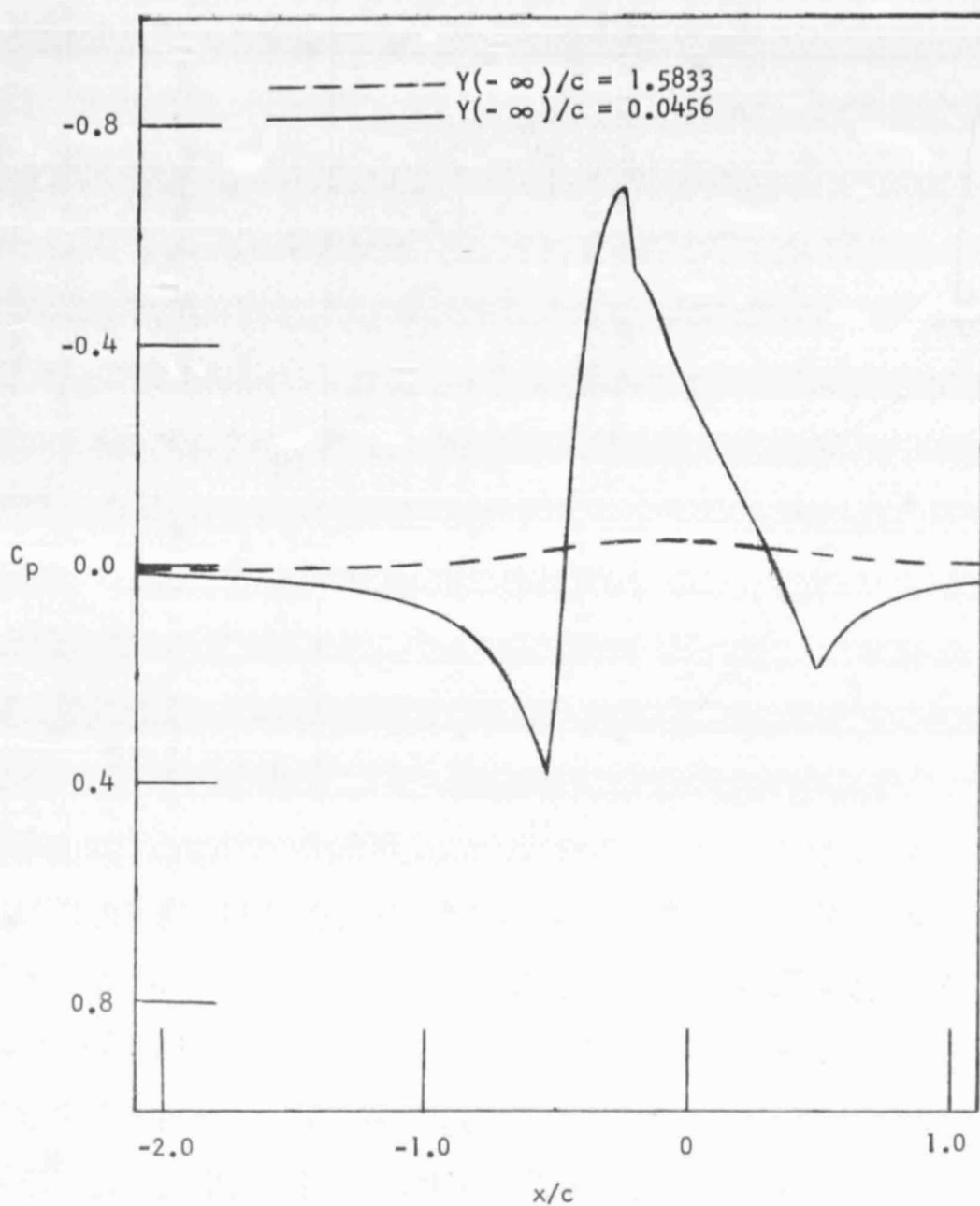


Figure 6.- Calculated pressure distributions along two streamlines.

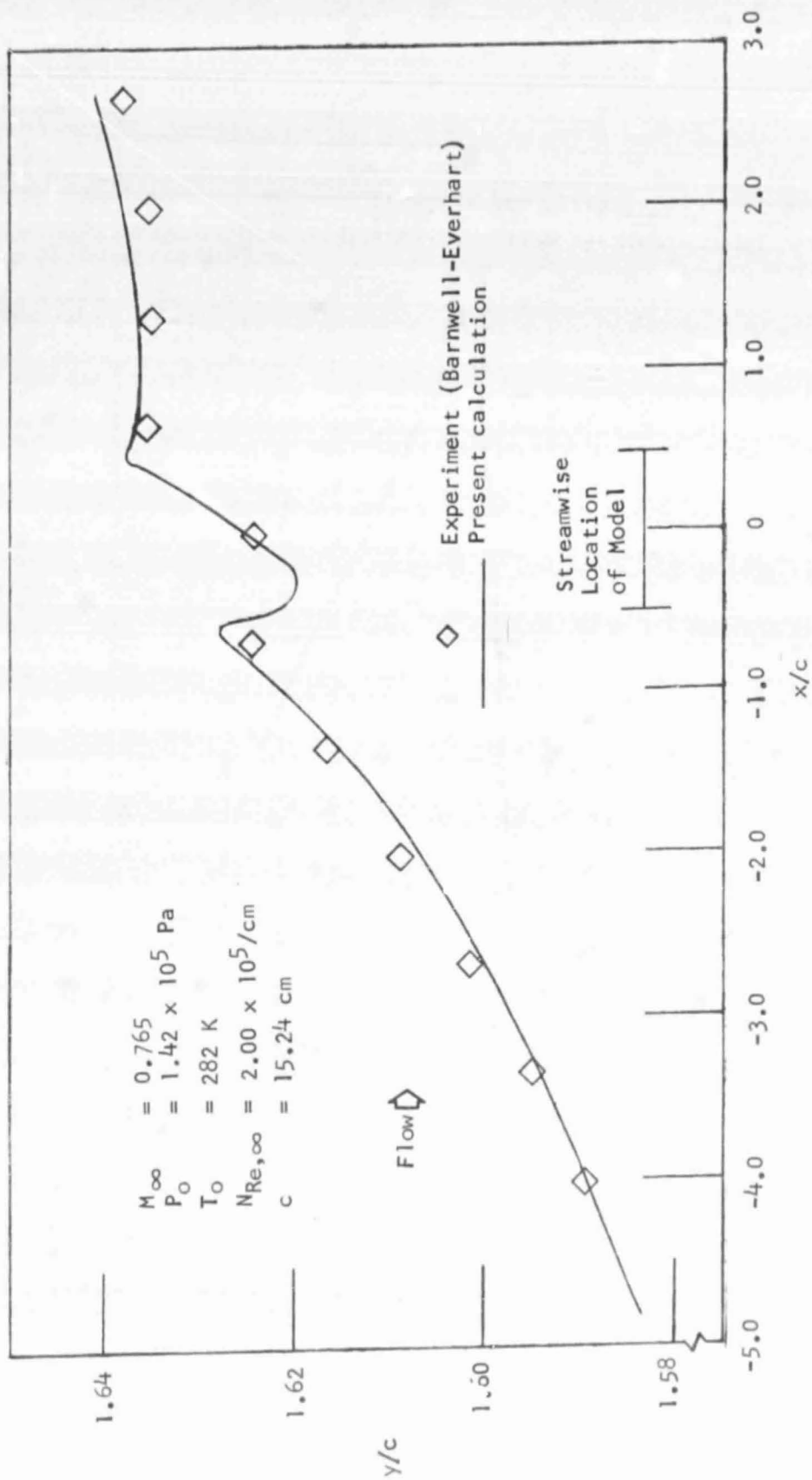


Figure 7.- Comparison of flexible nonporous wall shapes for an NACA-0012 airfoil test at zero lift in the NASA Langley 6- by 19-Inch Transonic Tunnel.

1. Report No. NASA TM 78641		2. Government Accession No.		3. Recipient's Catalog No.	
4. Title and Subtitle NUMERICAL DESIGN OF STREAMLINED TUNNEL WALLS FOR A TWO-DIMENSIONAL TRANSONIC TEST				5. Report Date April 1978	
				6. Performing Organization Code	
7. Author(s) Perry A. Newman and E. Clay Anderson				8. Performing Organization Report No.	
				10. Work Unit No. 516-53-03-14	
9. Performing Organization Name and Address NASA Langley Research Center Hampton, Virginia				11. Contract or Grant No.	
				13. Type of Report and Period Covered Technical Memorandum	
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546				14. Sponsoring Agency Code	
15. Supplementary Notes Dr. E. Clay Anderson is a consultant for DCW Industries, Inc., and has performed the boundary-layer analyses for this study under NASA Contract NAS1-14517.					
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17. Key Words (Suggested by Author(s)) Transonic testing Streamlined tunnel Adaptive wall tunnel				18. Distribution Statement Unclassified - Unlimited	
19. Security Classif. (of this report) Unclassified		20. Security Classif. (of this page) Unclassified		22. Price* \$4.00	
		21. No. of Pages 22			